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ORBIT DETERMINATION COVARIANCE ANALYSES FOR THE PARKER SOLAR PROBE MISSION

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This paper details pre-launch navigation covariance analyses for the Parker Solar Probe mission. Baseline models and error assumptions are outlined. The results demonstrate how navigation will satisfy requirements and are used to define operational plans. A few sensitivities are identified and the accompanying investigations are described. Predicted state uncertainty results show that most requirements are met with substantial margin. Moreover, navigation sensitivities may be accommodated operationally and this has been incorporated into project planning. Detailed results are presented only for select launch dates, however twenty unique trajectories (one per launch opportunity) have been assessed.

INTRODUCTION

The Parker Solar Probe (PSP), formally known as Solar Probe Plus (SPP), is a unique and historic mission, exploring what may be the final frontier in the solar system, the Sun's outer atmosphere. PSP will repeatedly sample the near-Sun environment to enhance our fundamental understanding of the corona, and the structure, origin, and evolution of the solar wind.[1] The spacecraft is designed and built by APL with navigation provided by JPL. The project's first launch opportunity is on July 31, 2018.

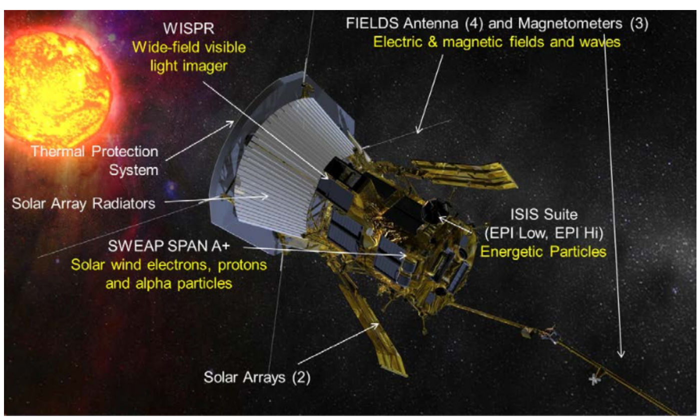

Figure 1. Parker Solar Probe Spacecraft[2]

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The reference trajectory design, depicted in Figure 2, involves launch aboard a Delta-IV Heavy and STAR 48BV upper-stage, followed by seven Venus gravity-assist flybys to gradually reduce perihelion from around 35.7 to 9.86 solar radii (R_S). The reference trajectory is entirely ballistic, consisting of 24 solar orbits over a mission duration of seven years.[2] A primary launch period of

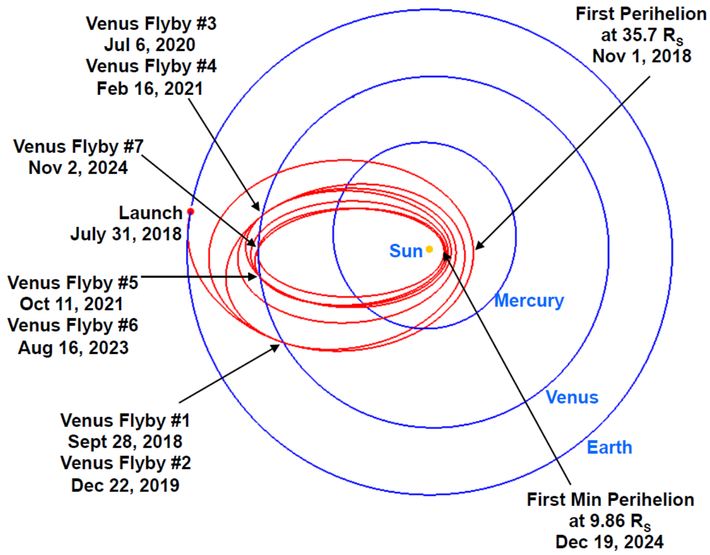

Figure 2. Reference Trajectory

20-days permits daily launch opportunities, and a similar backup launch period is available in 2019. More than 40 trajectory correction maneuvers (TCM) are scheduled to maintain the trajectory. The APL mission design (MD) team and the JPL navigation (NAV) team will converge upon a single TCM design, and therefore considerable effort has been undertaken to minimize dynamic modeling differences between the teams.

This paper focuses on orbit determination (OD). A concurrent paper covers flight path control results, including the assessments of statistical ΔV consumption.[3] Some important requirements levied against navigation are summarized in Table 1. Spacecraft position uncertainty below 1/4 AU and range-rate and range-rate-rate errors for DSN are the most stringent of these; however, the former is far more consequential since it derives from spacecraft pointing.

The environment and aspects of the mission design present new challenges for JPL navigation (NAV), particularly in terms of modeling the dynamics. Previously, Jones et al.[4] quantified and detailed various perturbations and associated errors which are unique for the mission. The effects include aberration of incoming solar radiation, drag from solar wind, solar limb darkening, and charged particle phase noise. However, most of these models are not required to capture the dominant sources of uncertainty necessary for covariance analysis. This paper summarizes current best estimate dynamics and the sources of uncertainty in these models as applied to the most recent trajectory cycle covariance analyses.

Table 1. Navigation Requirements

MDNR-22	$\Delta V99$ to not exceed 170 m/s
MDNR-24	Reconstructed heliocentric position errors no greater than 1200 km, 3-σ for $\leq$ 1/4 AU
MDNR-25	Predicted heliocentric position errors no greater than 1200 km, 3-σ for solar ranges $\leq$ 1/4 AU, delivered no later that 48 hr before last possible uplink
MDNR-26	Predicted heliocentric position errors no greater than 8500 km, 3-σ in any direction for solar range $>$ 1/4 AU and greater than 5 days from Venus encounter
MDNR-27	Predicted heliocentric velocity of 0.1 km/s, 3-σ in any direction
MDNR-74	Predicted Earth-line position $\leq$ 4 arcmin, 3-σ within 5 days of a Venus encounter
MDNR-73	Predicted Earth range, range rate, rate of range rate for DSN tracking
MDNR-75	Predicted Earth-line range errors no greater than 10,000 km, 3-σ for distance $>$ 1/4 AU

BASELINE COVARIANCE ANALYSIS

The primary sources of navigation uncertainty are associated with autonomous momentum desaturation maneuvers (on unbalanced thrusters) and solar and thermal radiation pressure. OD must also tolerate some long duration tracking gaps, where communication is not permitted due to low Sun-Earth-Probe (SEP) angle or when the thermal protection system (TPS) blocks the antenna. Moreover, measurements must be properly weighted as a function of SEP and declination (for ΔDOR).

A baseline covariance analysis is performed for each of the 20 unique reference trajectories corresponding to each opportunity in the primary launch period. The uniqueness of the reference missions is important since the timing of NAV relevant events (e.g. flybys, TCMs, tracking schedules and gaps) varies. The purpose of the study is to demonstrate requirement verification and to provide a notional data cutoff (DCO) schedule, for each reference mission. The DCO schedule determines when the spacecraft ephemeris is updated during operations (on board and with the DSN).

Nominal Trajectory

Covariance analysis is concerned with knowledge and variations about a nominal trajectory. MD and NAV integrated trajectories will not exactly match due to differences in dynamic modeling, integrators, and software. Nominal trajectories used for the analysis here are generated by NAV to achieve the same Venus B-plane targets as MD (as well as solar distance on the final three perihelia). Some deterministic ΔV is required to achieve the targets, but the so-called match ΔV is small. Table 2 lists the total match ΔV for open, middle, and close of the launch period. Figure 3 illustrates the trajectory differences between MD and NAV for the July-31 mission date.

Table 2. Deterministic Match ΔV in NAV Nominal Trajectories

Date	ΔV, m/s
July-31	4.30
August-09	4.90
August-19	4.43

The trajectories agree at each of the Venus flybys, and the differences are relatively small. It is assumed that the current small match ΔV will get even smaller as the two teams converge towards operational modeling.

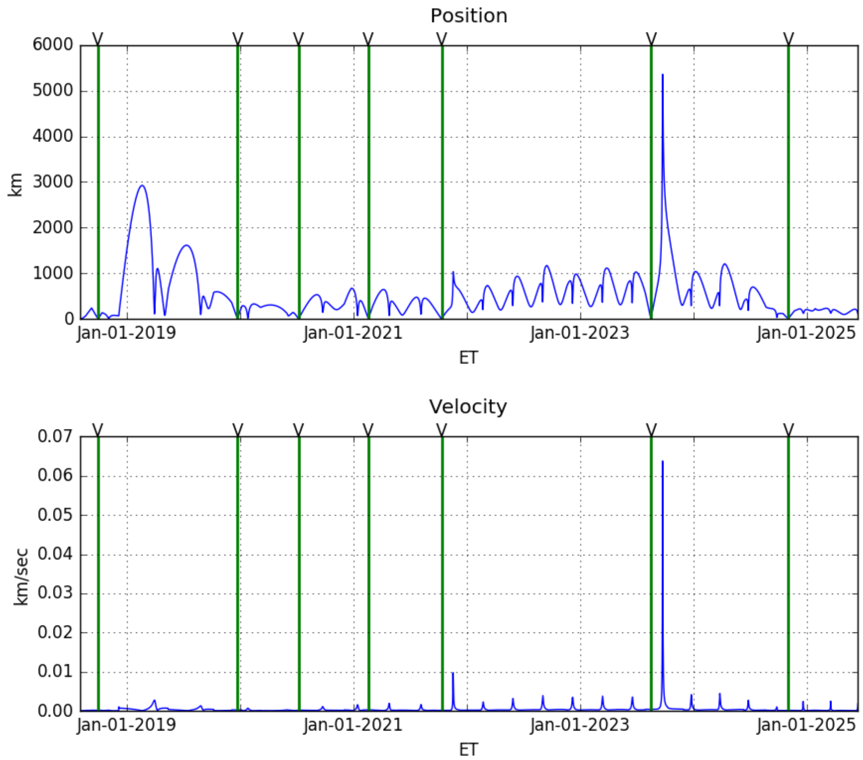

Figure 3. MD and NAV Reference Trajectory Differences for July-31 launch

Force modeling

PSP is primarily perturbed by solar radiation pressure (SRP) and momentum desaturation maneuvers (desats). The spacecraft consists of a prominent thermal protection system (TPS), two primary solar panels, two secondary panels, an upper bus structure housing the radiators (cooling system), and a lower spacecraft bus structure. For most of operations the TPS is directed to the Sun, and the panels rotate (by a flap angle) as a function of solar distance in order to maintain the delicate balance between heat and power. For this study, the TPS is permanently directed to the Sun. Figure 4 depicts the solar panel array configuration at the minimum solar distance, where each wing (side) consists of a primary and a secondary panel.

Navigation uses flat plate shape elements to model the deterministic SRP force. The force nominally has only a radial component. Various non-radial accelerations are treated as error sources (e.g. non-radial acceleration for due to a TPS pointing bias). For this study, the TPS and four solar panel elements are modeled as flat plates. The general force contribution from a flat plate[5] may be written as:

$$\mathbf{f}_{\text{plate}} = \frac{C\ A}{r^2}\,\bar{\mathbf{f}} = \frac{C\ A}{r^2}\left[(2\mu - 1)\cos\alpha\ \hat{u}_r - (2\nu + 4\mu\cos\alpha)\cos\alpha\ \hat{u}_n\right] \tag{1}$$

Where C is a constant with units of force, r the solar distance, and A is the effective area. Also, $\hat{u}_r$ is the incoming light unit direction, $\hat{u}_n$ is the unit direction normal to the plate, α is the angle between $\hat{u}_n$ and $\hat{u}_r$, and μ and ν are specular and diffuse reflectivity coefficients, respectively. During operations the incoming light will not exactly coincide with the ephemeris Sun direction, because of aberration of light effect. However, this small difference is reasonably neglected for this study since the attitude control system will be accounting for aberration.

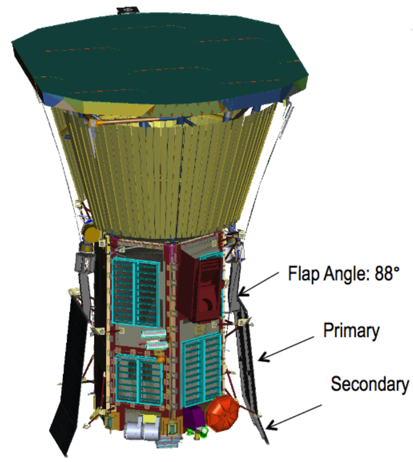

Figure 4. PSP spacecraft depicting solar panel array geometry.

Due to the significant amount of thermal emission (re-radiation out the front), the TPS is best approximated as a perfectly diffuse reflector ($\mu = 0$ and $\nu = 1/3$). This differs from the body's optical properties. As detailed by Jones et al.[4], navigation approximates the solar panel directions and effective areas (to account for shadowing) as a discrete time-series. The time-series are proportional to solar distance, with the flap angle approximated as a power series in solar distance. The panels are assumed to be symmetric during the study such that the non-radial force contribution is zero.

Below 1/4 AU desats occur autonomously, but a notional schedule is adopted for deterministic force modeling. The uncertainty in the timing (and number) of these events is a significant contributor to overall OD error. The best estimate for the residual ΔV vector (each event) is $[1.5,\ 0.0,\ 9.6]$ mm/sec. The complete set of deterministic force models assumed are listed in Table 3.

Table 3. Reference trajectory force models

Model	Details
Planetary ephemeris and GM	DE430
Earth gravity	GGM03C
Venus gravity	point mass
SRP	TPS and solar panels (with shadowing)
Attitude	Sun-pointed
Solar panel normal directions	Interpolated as power series in solar distance
Notional momentum desaturation impulses	Per G&C schedule

Error modeling

Because the TPS is assumed Sun-pointed and the attitude is symmetric with respect to the Sun-direction, the deterministic SRP force is entirely radial. The uncertainty in TPS area and reflective properties dominates the radial component of SRP error. This is accounted for with a stochastic scale factor applied to the entire spacecraft shape (which consists of just the TPS to a first order approximation). In addition, some uncertainty must be added to account for non-radial force errors which arise from asymmetric thermal re-radiation, TPS alignment error and attitude error, asymmetric solar panels, etc. These uncertainties are aggregated into a non-physical S/C-bus element. This element permits arbitrary accelerations that vary as the inverse square of solar distance. This element's force contribution is given by Eq. (2) (in the spacecraft body frame).

$$\mathbf{f}_{\text{sc-bus}} = \frac{C\ A_{\text{bus}}}{r^2} \begin{bmatrix} G_x \\ G_y \\ G_z \end{bmatrix} \tag{2}$$

Where the G coefficients are factors estimated with appropriate area factor A_{bus}. The area is set to 0.07% of the TPS, based on analysis bounding the various sources of non-radial SRP acceleration[6], such that an expected 1-σ acceleration yields $G_x = G_y = 1$.

Table 4. Covariance study error models

Estimated Parameters

Error source	**1-σ uncertainty**	**Update time**	**Correlation time**	**Comments**
Initial position Initial velocity	1000 km 1 km/sec			Epoch 30-days prior to previous encounter
Solar pressure scale factor	10%	3 days	7 days	
S/C bus shape model (GX, GY)	Equivalent to 0.07% of TPS area in both X and Y directions	3 days	7 days	Internal memo[7]
per-pass range biases	2 m			
TCM execution errors	G&C error model and statistical ΔV results			Correlated covariance
Momentum dump frequency error ΔV	$[0.5,\ 0.6,\ 3.2]$ mm/sec	Per event	10 days	

Considered Parameters

Error source	**1-σ uncertainty**	**Comments**
GMs: Venus, Earth, Moon, Sun, Mercury	DE430	IPN progress report[8]
Venus and Earth ephemeris		Internal memo[9]
Station locations	Full covariance	
Quasar locations	1 nrad	
Earth orientation X,Y	1.5E-8	
UT1	3.0E-4	
Ionosphere: day/night	75 cm/15 cm	
Troposphere: wet/dry	4 cm/1 cm	

The greatest error from desats results from the uncertainty in the frequency of autonomous events inside 1/4 AU, rather than the single event execution error. The frequency is primarily a function of Cp-Cg offset, and the center-of-pressure (Cp) is not very well known. For this analysis the frequency error is assumed to be the resulting frequency difference between nominal and worst-case Cp-Cg offsets (3-σ) as determined by G&C team Monte Carlo. This uncertainty is applied as a biased (or correlated) ΔV offset upon all notional desats in a given arc. This level of ΔV uncertainty is overly conservative later in the mission, because once navigation and G&C teams are able to observe the number of desats for some number of perihelion passes, the uncertainty can be reduced. A complete set of estimated and considered error sources and values are presented in Table 4.

The primary reason for using ”considered” error sources is to identify any models which cause problems if ignored. These uncertainties are assumed to be unimproved by the filter, and therefore constitute systematic modeling error that is absorbed into mapped state error and statistical ΔV.

Tracking data

A notional tracking data schedule is based on the rules outlined in Table 5. The frequency of data is however subject to revision subject to NAV analysis. Whenever possible alternating North-South (Goldstone-Canberra) and East-West (Goldstone-Madrid) baseline ΔDOR measurements are requested and therefore assumed. However, the PSP trajectory is often at low declination which results in unavailable or degraded ΔDOR from Madrid. During such periods, the North-South baseline is preferentially utilized.

Table 5. Tracking Schedule Assumptions

Doppler and Range

From	To	Type of Support
Launch	Launch + 1 weeks	continuous
Launch + 1 week	Launch + 2 weeks	16-hours per day
Venus − 5 weeks	V − 1 week	five 10-hour passes per week
Venus − 1 week	V + 1 week	one 10-hour pass per day
————	————	three 8-hour passes per week

ΔDOR

From	To	Type of Support
Venus − 3 weeks	Venus + 2 weeks	two tracks per week
————	————	one track per week

Coherent two-way data is entirely unavailable for significant portions of the mission including when SEP is below 2.1 degrees, when attitude causes the TPS to block the antenna, and when the solar range is below 10 R_S. Range and ΔDOR tracking are further restricted on the basis of available bandwidth. For example, the ranging link is periodically closed because of lower signal-to-noise ratio. These outages are collectively referred to as tracking gaps, and the gaps are launch date specific. NAV reads a spreadsheet of gaps provide by the Telecomm subsystem team and then applies this to the covariance study simulation. Table 6 gives a summary of the total simulated tracking data for the July 31st launch date. Note, that ranging data is reduced 33% relative to Doppler and there are many more North-South baseline ΔDOR measurements because of time spent at low declination.

Table 6. Tracking data summary for July 31, 2018 launch date

	Number of tracks
Doppler	1090
Range	732
North-South baseline ΔDOR	119
East-West baseline ΔDOR	56

Two-way data is mostly X-band (up and down), but X-up/Ka-down is used by NAV during times of high-bandwidth science downlink. For this study, data is conservatively weighted at the expected noise level of X-band data. Data must also be de-weighted (from nominal) during times of increased noise. Weighting assumptions are given in Table 7. At low SEP, solar plasma will cause signal phase scintillation. To account for increased noise and unreliable measurements, the data is deweighted. For this study, the discretized table of scale factor weights in Table 8 are applied. The weights are derived from a power-law fit of SEP and solar range using data from a variety of sources.[10,11] During operations a more sophisticated weighting algorithm and/or a whitening (de-correlation) algorithm will be utilized.[4,12]

Table 7. Tracking data weighting, 1-σ

Data type	Nominal weight	Deweighting
Doppler	0.1 mm/sec	De-weighted at low-SEP angle using a power-law fit
Range	10 m	De-weighted at low-SEP angle using a power-law fit
ΔDOR	0.06 ns	East-West baseline at 0.2 ns for declination below -15 degrees. East-West baseline not permitted for declination below -25 degrees.

Table 8. Radiometric low-SEP deweighting schedule

Scale factor	SEP angle, deg	
	from	to
1	–	> 31
2	31	17
5	17	8.3
10	8.3	4.7
20	4.7	2.8
100	2.8	2.1

Ephemeris update assumptions

The most stringent NAV requirements involve the accuracy of the predicted spacecraft ephemeris. How large the predicted error becomes is directly tied to how often (and when) OD performs an update. The ephemeris updates are constrained by the time it takes to turn-around a solution (and uplink it) and by the availability of a relatively high-bandwidth uplink opportunity. The following relate the DCO time to the time when the corresponding OD is said to be active (uplinked to the spacecraft):

- Turn-around time to perform and deliver OD is 12-hours.
- DSN receives updated ephemeris immediately (12-hours after DCO).
- The DSN is updated at each spacecraft ephemeris update (no less or more).
- The time from DCO until the ephemeris is uploaded on the spacecraft is nominally 5-days (and 3-days minimum). This includes minimum of 2-days for spacecraft team to prepare and test the upload.

Additionally, no ephemeris upload is permitted inside 1/4 AU. The following rules for DCO placement, were devised by NAV, and are sufficient to meet requirements across mission dates:

- TCM-01 minus 3 days
- TCM-XX minus 5 days
- 1/4 AU entry minus 7 days (unless in gap)
- Venus plus 2 days (with minimum 3-day turn-around time)
- Maximum duration between subsequent DCOs of 30-days (unless gap prevents)
- No DCOs allowed in the time intervals: 1/4 AU entry minus 3 days through 1/4 AU exit minus 3 days

For the July 31 2018 launch date this resulted in 119 ephemeris uploads. For some mission dates DCOs need to be manually placed around long tracking gaps, but all other dates are within ± 5 uploads of 119.

Filter assumptions

Each reference mission trajectory is split into arcs, each covering two encounters (Venus or solar periapsis). This is necessary since integrating through multiple gravity assist flybys introduces large numerical errors into the simulation. For each arc, the initial state is acquired by back integrating from the reference trajectory target condition (i.e. B-plane targets for Venus) to the initial time. The arc is then forward integrated past the encounter, to just before the first post-encounter TCM. Table 9 lists the arcs with start and end times relative to events. Each arc covers a single targeted encounter and at least 5-days a Venus encounter (covering MDNR-74 which applies $\pm$ 5-days about Venus).*

Table 9. Covariance Study Arcs

Name	Initial epoch	Final epoch
V1	Launch	V1 plus 12 days
V2	V1 minus 30-days	V2 plus 13 days
V3	V2 minus 30-days	V3 plus 9 days
V4	V3 minus 30-days	V4 plus 16 days
V5	V4 minus 30-days	V5 plus 55 days
V6	V5 minus 30-days	V6 plus 53 days
V7	V6 minus 30-days	V7 plus 17 days
P22	V7 minus 30-days	P22 plus 38 days
P23	P22 minus 30-days	P23 plus 28 days
P24	P23 minus 30-days	P24 plus 60 days

At each DCO a filter solution is computed and mapped forward in time until the next DCO ephemeris upload time (typically 5-days after DCO). The first mapping time is coincident with the ephemeris upload time, except for the DSN where it is 12-hours after the DCO. Maneuver execution errors are included only for first upcoming TCM.

*P22 is the twenty second perihelion, and the first of the three primary science passes.

BASELINE RESULTS

The predicted position error requirements (MDNR-25,26) are verified together by plotting the errors. Figure 5 verifies the requirements being met relatively easily for the July-31 launch date covering launch through the Venus-3 encounter.[†] Figure 6 is an analogous plot but covers from Venus-6 encounter to the end of mission.

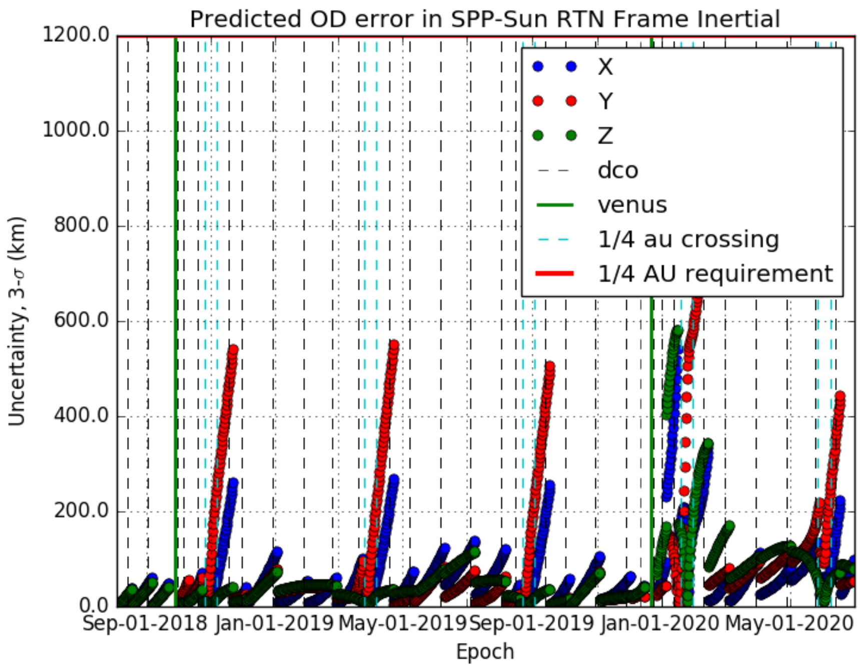

Figure 5. Predicted position errors for July-31 launch date mission, Venus 1 through Venus 3

The uncertainty does exceed the 1200 km requirement limit in a couple places, but since these times are outside 1/4 AU the requirements are indeed satisfied (uncertainty need only remain below 8500 km outside 1/4 AU). The point of maximum error inside 1/4 AU occurs between V6 and V7, and coincides with the longest tracking gap. NAV has studied (and continues to study) sensitivities associated with missing and/or degraded data around this time. MDNR-24 is implicitly satisfied since reconstructed error is strictly less than predicted error.

The predicted velocity (MDNR-27) and predicted angular uncertainty near Venus (MDNR-74) requirements are verified with similar plots. Representative examples are found in Figures 7-8. For all mission dates, these two requirements are met with significant margin.

The predicted spacecraft ephemeris requirement for the DSN (MDNR-73) is challenging to evaluate since they only apply to times of scheduled tracking. Moreover, the consequences of violation entails some difficulty acquiring lock or at worst some data loss. The range and angular uncertainty components of the requirement are satisfied very easily. The range-rate and range-rate-rate components are exceeded at Venus flybys and near many of the 1/4 AU crossings. Figure 9 is an example of the range-rate-rate violation, which has (unsurprisingly) large error near the Sun where perturbations (SRP and desats) are large and fairly uncertain. If operational tracking is scheduled

[†] In the plots, X corresponds to radial, Y to along-track, and Z to normal.

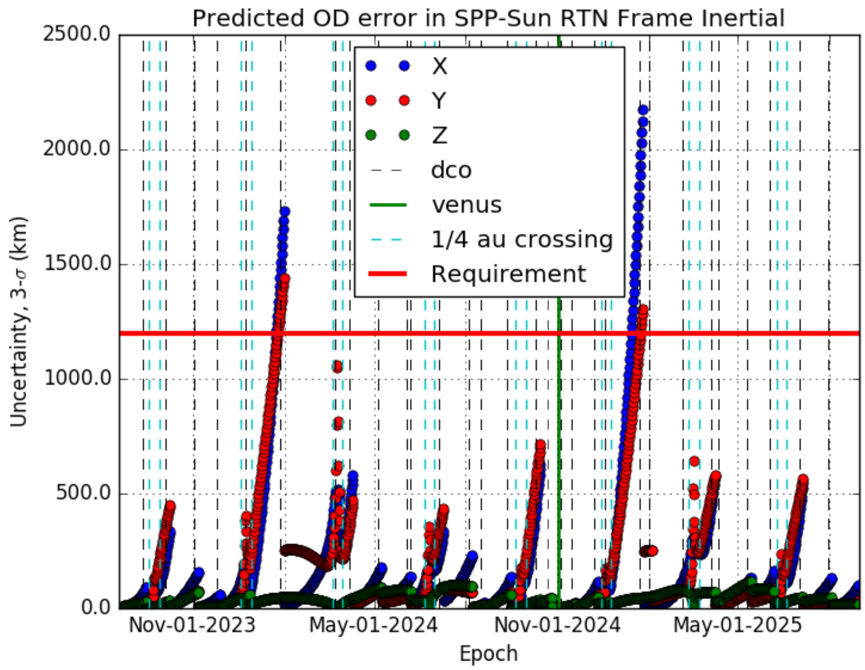

Figure 6. Predicted position errors for July-31 launch date mission, Venus 6 through Perhelion 24

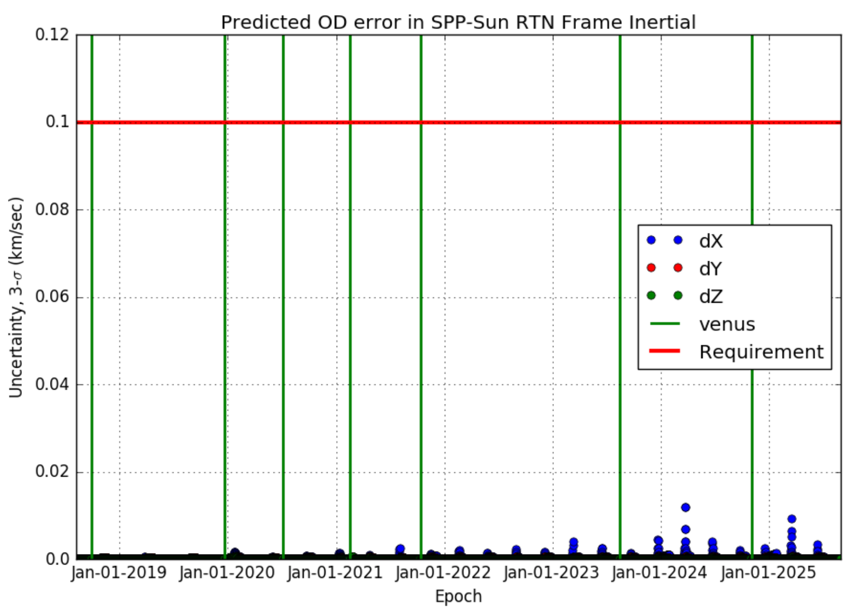

Figure 7. Predicted velocity errors for July-31 launch date mission

during times of violation, the project must only accept the possibility of longer than usual time to acquire lock and the potential for reduced data quality. Nevertheless, these violations are examined via sensitivity analysis to demonstrate that even extremely conservative data loss is not detrimental to meeting the mission critical navigation requirements. Plots like Figures 5-9 (and others) were generated for all 20 mission dates, and show requirements are met across the entire launch period.

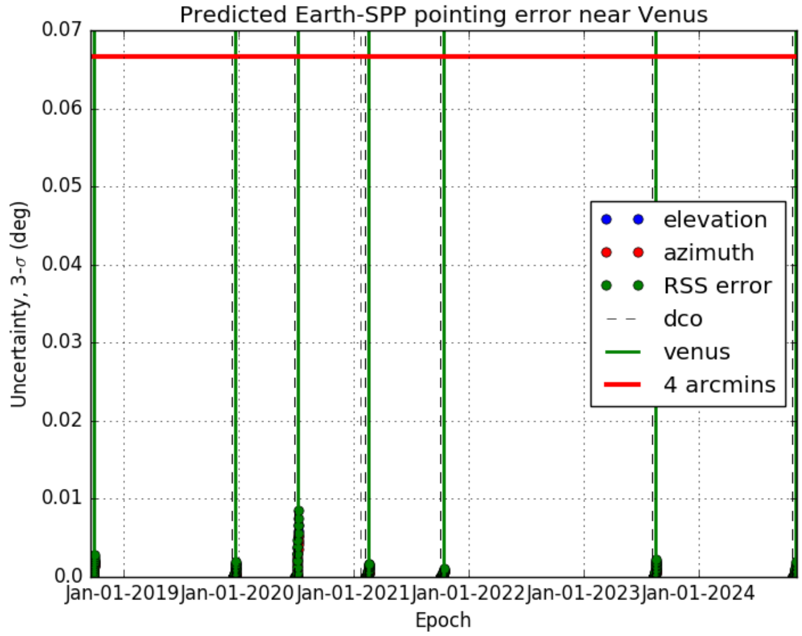

Figure 8. Predicted Earth-PSP direction errors $\pm$ 5-days of Venus for July-31 launch date mission

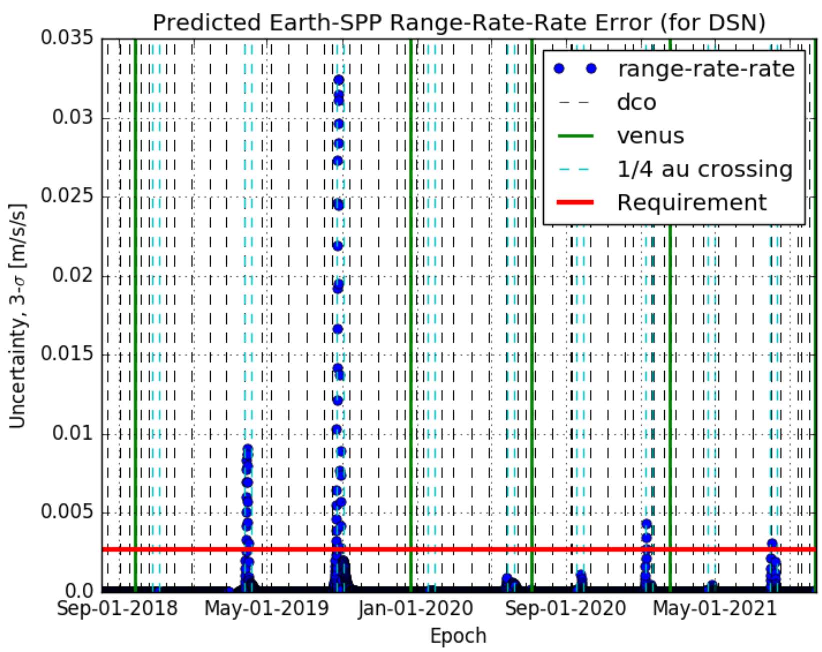

Figure 9. Predicted range-rate-rate error for July-31 launch date mission, Launch through Venus 6

SENSITIVITY ANALYSES

A number of sensitivity cases are considered to assess margins in meeting requirements. For example, if tracking data is lost or degraded, or if environmental perturbations are off-nominal (e.g. a coronal mass ejection event). PSP is at times sensitive to the number of ΔDOR measurements, particularly near Venus. Nearly adjacent tracking gaps also introduce operational strain. These times are identified to ensure the criticality is well known and that sufficient personnel are available. The following sensitivity cases are presented:

1. Geometrical sensitivity after Venus 2 flyby
2. Missed ephemeris update between long tracking gaps around P18 and P19 because of degraded/lost data and/or un-prioritized two-way scheduling.
3. Consequence of lost data because of large errors in predicted range-rate and range-rate-rate near Venus and near 1/4 AU.

Sensitivity post-Venus 2 flyby

This sensitivity applies to most mission dates, but is particularly prominent for the August-10 launch date. It was discovered that the PSP post-flyby velocity vector is nearly aligned with the Earth direction at this time. Without sufficient ΔDOR measurements the predicted OD requirement can be violated. The dominant error source is the planetary ephemeris consider parameters. Because of the geometry these uncertainties are indistinguishable from state error. This will be addressed in operations by estimating the Earth/Venus ephemeris (rather than considering).[‡] Estimating eliminates the sensitivity, however, ΔDOR measurements will be given elevated priority around this time. The "considered" parameters worked as intended here, because they alerted NAV to the potential for a problem if the errors were left uncorrected.

Missing an ephemeris update between P18 and P19

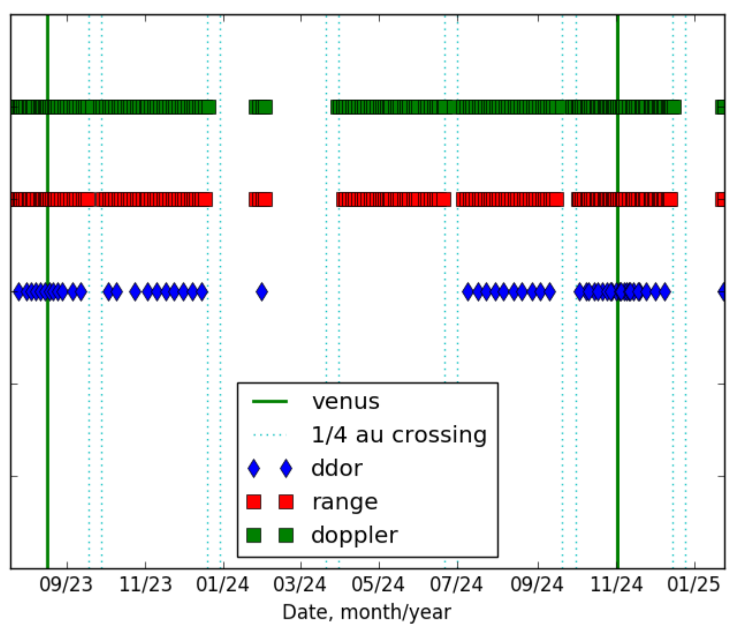

Figure 10. Simulated tracking data between Venus 6 and 7

[‡]The planetary ephemeris apriori will likely not be updated for later batches.

There are two long (nearly adjacent) tracking gaps between perihelion 18 and 19. The gaps are evident in Figure 10, which shows simulated tracking between Venus 6 and 7. This point of the mission is particularly sensitive for navigation, and the largest state errors (inside 1/4 AU) occur here. Moreover, there is only around a week between the two gaps, and the SEP is quite low. Figure 11 illustrates how the predicted OD would blow-up if the upload opportunity (between the gaps) was missed. Adequate personnel and tracking schedule priority is critical to avoid a spacecraft safe-mode resulting from a missed onboard ephemeris update. Thankfully, very little data (despite being degraded) is needed to keep the error within requirements. In fact, a single 8-hour tracking pass should suffice.

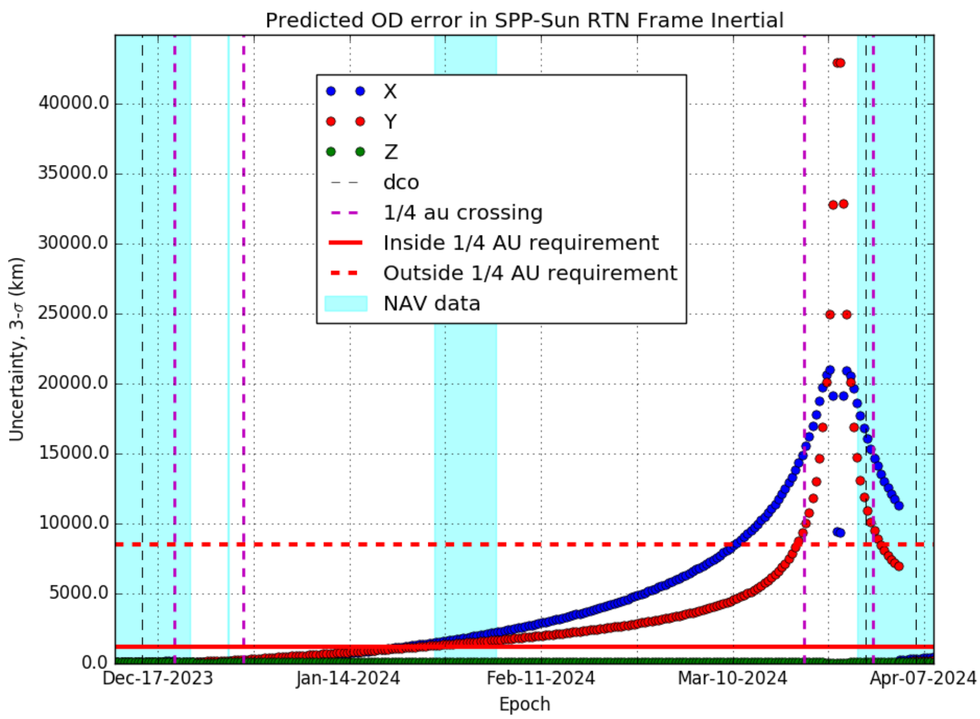

Figure 11. Predicted position error without an upload between P18/P19 tracking gaps

Tracking data loss during times of large range-rate and range-rate-rate uncertainty

As shown in Figure 9, errors in the derivatives of range can get large at flybys and near the Sun. NAV very conservatively assessed what would happen if all of this data (whenever a violation occurs) is lost. Doing so resulted in predicted errors which were larger overall (relative to the baseline analysis), but still well within requirements. Furthermore, the ΔV penalty associated with the lost data was found to be statistically insignificant.[3]

CONCLUSIONS

The Parker Solar Probe navigation team has demonstrated that all orbit determination requirements can be met for all 20 unique mission dates in the launch period. The covariance analysis used in this demonstration will guide the efforts into the final trajectory cycle analysis prior to launch.

A number of sensitivities have been identified and are readily addressed. This mission is venturing into a yet unexperienced regime of space, which has necessitated a fresh look at certain aspects of high-precision orbit determination processes in place at JPL. There will undoubtedly be surprises during operations, but the level of conservatism carried throughout this analysis should provide sufficient margin to meet all goals and deliver exceptional navigation performance.

ACKNOWLEDGMENTS

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REFERENCES

[1] N. Fox, M. Velli, S. Bale, R. Decker, A. Driesman, R. Howard, J. Kasper, J. Kinnison, M. Kusterer, D. Lario, M. Lockwood, D. McComas, N. Raouafi, and A. Szabo, "The Solar Probe Plus Mission: Humanity's First Visit to Our Star," *Space Science Reviews*, 2015, pp. 1–42, 10.1007/s11214-015-0211-6.

[2] Y. Guo, J. McAdams, M. Ozimek, and W.-J. Shyong, "Solar Probe Plus Mission Design Overview and Mission Profile," *24th International Symposium on Space Flight Dynamics (ISSFD), Laurel, Maryland*, May 5-9 2014.

[3] P. Valerino, P. Thompson, D. Jones, T. Goodson, M. Chung, and N. Mottinger, "Flight Path Control Analysis for Parker Solar Probe," to appear in proceedings of the 2017 AAS/AIAA Astrodynamics Specialists Conference.

[4] D. R. Jones, T. Goodson, P. Thompson, P. Valerino, and J. Williams, "Solar Probe Plus: Unique Navigation Modeling Challenges," *AIAA/AAS Astrodynamics Specialist Conference*, Long Beach, CA, September 2016, 10.2514/6.2016-5425.

[5] R. M. Georgevic, "Mathematical model of the solar radiation force and torques acting on the components of a spacecraft," tech. rep., JPL Technical Memorandum 33-494 NASA-CR-123361, Oct 1971.

[6] D. R. Jones, "Solar Probe Plus (SPP) Non-Radial Accelerations," *JPL IOM 392A-16-001 (internal document)*, January 2016.

[7] D. R. Jones, "Solar Probe Plus (SPP) non-gravitational force model description," *JPL IOM 392A-17-002 (internal document)*, 2017.

[8] W. Folkner, J. Williams, D. Boggs, R. Park, and P. Kuchynka, "The Planetary and Lunar Ephemerides DE430 and DE431," *IPN Progress Report 42-196*, February 2014.

[9] R. Park and W. Folkner, "Uncertainty in the orbits of Venus and Earth barycenter for Akatsuki (Planet-C) planning," *JPL IOM 392R-15-003 (internal document)*, 2015.

[10] W. M. Folkner, "Effect of Uncalibrated Charged Particles on Doppler Tracking," *JPL IOM 335.1-94-005 (internal document)*, March 1994.

[11] J. D. Anderson, P. A. Laing, E. L. Lau, A. S. Liu, M. M. Nieto, and S. G. Turyshev, "Study of the anomalous acceleration of Pioneer 10 and 11," *Phys. Rev. D*, Vol. 65, Apr 2002, p. 082004.

[12] R. Mackenzie and W. M. Folkner, "Applying Appropriate Weights to Doppler data," *JPL IOM 343J-06-034 (internal document)*, October 2006.